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FEASIBILITY STUDY OF A CIRCUMLUNAR PHOTOGRAPHIC EXPERIMENT

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SUMMARY

A study has been made to investigate the feasibility of a high-resolution, lunar-surface photographic experiment, with the use of a circumlunar trajectory and with recovery of the film on return to the surface of the earth. Particular attention has been given to procedures for obtaining high-resolution photographs of the lunar surface, for returning the undeveloped film to the earth, and for recovering the data package on completion of such a mission. As an example of a typical existing vehicle which could be used for such an experiment, the characteristics of the Ranger spacecraft have been used in applicable portions of the study.

Consideration has been given to trajectory analysis for minimizing the dispersion area on return to the surface of the earth, design of the optical system and photographic-axis attitude orientation, radiation protection of the film package, reentry heating protection and recovery of the package, design details of incorporation of the experiment on the Ranger spacecraft, and additional experimental results and scientific information to be obtained from such a mission. These considerations demonstrate the technical feasibility and the scientific usefulness of such a mission.

INTRODUCTION

A study has been made to investigate the feasibility of a high-resolution, lunar-surface photographic experiment, with the use of a circumlunar trajectory and with recovery of the film on return to the surface of the earth. Particular attention has been given to procedures for obtaining high-resolution photographs of the lunar surface, for returning the undeveloped film to the earth, and for recovering the data package on completion of such a mission. As an example of a typical existing vehicle which could be used for such an experiment, the characteristics of the Ranger spacecraft have been used in applicable portions of the study.

The primary objective of such a mission is to obtain lunar-surface photographs and vidicon (television) pictures for determination of surface features and formations over a wide area of the eastern limb down to a resolution of 80 feet, for determination of altitudes and slopes of the surface features, for photometric analysis of the surface structure to this scale, and for determination of the figure of the moon along the earth-moon axis. Additional objectives include the possibility of more precise determination of the masses of the earth and moon from analysis of the tracking data and a study of phenomena associated with the effect of proximity of moon and vehicle on communications and with loss of line-of-sight communications as the vehicle passes behind the moon. In addition to the results previously mentioned, the mission offers a demonstration of circumlunar flight with recovery on return to the earth, a communications and tracking exercise, some data on reentry heating at parabolic velocity, and photographic data for selection and features of possible landing sites. It is also anticipated that some of the techniques and operational concepts used in the photographic experiment could be of direct application to a lunar sample return mission.

The primary results would consist of photographs of the lunar surface taken during the lunar-approach phase of the single-pass circumlunar trajectory. At the third-quarter lighting condition, photographs of the entire eastern limb would be obtained, including the portion of this limb which has not yet been photographed. The photographs would be taken from three different inertial attitudes and thus would provide data for detailed analysis of surface features. Two magnification levels would be used, the first giving views of the whole lunar disk, with resolution down to 2,000 feet, and the second giving views of areas approximately 100 miles in diameter, to be photographed with a resolution down to approximately 80 feet. Photographs with this latter resolution could provide important data in the intermediate range between present earth-based photographs and the high resolution, but very limited field, vidicon pictures to be taken on impact missions. In addition to the film cameras, it would be useful to include the vidicon picture system already designed for use in the Ranger spacecraft.

Circumlunar trajectory studies have led to the design of typical nominal trajectories utilizing proper trajectory energy and minimum approach distance to the moon and a nearly vertical reentry on return to earth, for which the recovery area on return to the surface of the earth is relatively insensitive to errors remaining after application of a single midcourse correction. For the accuracies obtainable with the single midcourse correction of the Ranger spacecraft, and with standard uncertainties in the masses of the earth and moon, the estimated dispersion of the recovery point is approximately 730 statute miles east and west and 314 statute miles north and south from the nominal point. Within this dispersion area, the actual recovery point should be predictable within a few tens of miles from tracking data obtained up to 2 days before reentry, and with considerably greater accuracy than this from data obtained to within a few hours of reentry.

As part of the feasibility study, consideration has been given to design details of incorporation of the experiment on the Ranger spacecraft, design of the optical system and photographic-axis attitude orientation, radiation protection of the film package, reentry heating protection and recovery of the reentry package, and additional experimental results and scientific information to be obtained from such a mission.

This report contains more detailed discussions of the results of analyses generated in the course of the feasibility study and of the scientific results which would be obtained. Although some of the results and design considerations contained herein are of a preliminary nature and, as such, are subject to changes to be dictated by a thorough design study, it is believed that the information presented is sufficient to demonstrate the technical feasibility and scientific usefulness of such an experiment.

RESULTS AND DISCUSSION OF FEASIBILITY STUDY

Description of Ranger Vehicle

The objective of the Ranger program is the development of a spacecraft and system for performing scientific experiments in earth-moon space and for making a landing on the moon. A basic spacecraft, designated as the bus, will be used for all flights, with the addition of a landing capsule for the lunar impact shots. The spacecraft will be injected into orbit by the Atlas-Agena B vehicle. Ranger spacecraft design information required in the formulation of this study has been obtained from Jet Propulsion Laboratory (JPL) Space Programs Summaries for 1960 and 1961.

The first two flights of the Ranger program, designated as RA-1 and RA-2, will be highly elliptical earth orbits with apogee of about three times the earth-moon distance. Experiments to be contained in these flights are adequately described in reference 1 and will not be discussed here. The next three flights, designated as RA-3, RA-4, and RA-5 will be attempts to achieve a lunar impact with terminal velocity of the landing capsule of one hundred to several hundred feet per second. Experiments for these flights consist of a vidicon (television) photographic experiment for obtaining high-resolution pictures of a small part of the lunar surface in the vicinity of the impact area and a lunar radioactivity experiment, both of which are external to the (soft) landing capsule, and a seismometer experiment in the landing capsule.

The Ranger spacecraft is equipped with solar and earth sensors, three body-fixed rate gyros, an attitude control system, midcourse correction system, and, for the landing capsule, a retro-rocket system. Shortly after injection, ground command initiates the sun acquisition phase which results in the vehicle longitudinal axis (the roll axis) becoming oriented along the vehicle-sun line. Control torques for the attitude control maneuvers are obtained by gas jets. After orientation of the roll axis, earth acquisition is obtained by a roll maneuver about the longitudinal axis, accompanied by sweeping of the high-gain antenna (see fig. 1) to which are attached the earth sensors. After earth acquisition, the spacecraft orientation with respect to the sun and earth is maintained during the cruise phase.

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For the midcourse correction maneuver, included in missions RA-3 to RA-5, a premaneuver phase maintains spacecraft orientation at increased accuracy, then a midcourse orientation maneuver orients the body-fixed rocket motor in the proper direction for applying the required velocity correction. An autopilot maintains the spacecraft orientation during the correction maneuver, using jet vanes to give pitch, yaw, and roll control. The motor is ignited at a preset time, burns until the required velocity has been obtained, and is then shut off. After the midcourse correction, the sun and earth are reacquired and the vehicle is reoriented to the cruise orientation.

A similar procedure is employed for the terminal maneuver of RA-3, RA-4, and RA-5 in which the proper orientation is obtained for taking the vidicon pictures and for firing the retro-rocket.

The gross weight of the Ranger (RA-3) spacecraft is approximately 800 pounds. The JPL studies indicate that such a payload can be accommodated by the Atlas-Agena B vehicle, taking into account subnominal performance of the boost vehicle. Of the 800-pound gross weight, the landing capsule, including the retro-rocket system and payload, weighs 300 pounds.

For the considerations of the present study of the lunar photographic experiment, the landing capsule is eliminated. This results in a payload allowance for the photographic experiment and associated system of about 300 pounds. This 300-pound allowance is in addition to the attitude control system, communication system, midcourse correction system, power supply, and so forth which are already included in the basic spacecraft. Preliminary weight estimates indicate that a payload weight allowance of about 300 pounds is adequate for the proposed experiments.

Trajectory Analysis

The basic concepts involved in the design of a circumlunar trajectory for an unmanned mission differ somewhat from those involved in a manned mission. Primarily, these differences are the result of a relaxation of reentry restrictions, since for the unmanned mission it is no longer necessary to restrict the design to those trajectories having a rather limited reentry corridor. Thus, for the unmanned mission, it is possible to base the trajectory design on the results of an examination of the overall characteristics of circumlunar trajectories.

It was noted in reference 2 that certain circumlunar trajectories are characterized by the fact that the return perigee distance is relatively insensitive to changes in the distance of closest approach to the moon. The return perigee distance for which this occurs is dependent upon the energy, or velocity, of the trajectory. Thus, it would appear advantageous to design the trajectory to incorporate this inherent insensitivity. In addition, it would be desirable to choose the energy of the trajectory so that the return perigee is near the center of the earth. There are two advantages involved in making this choice. First, allowance can be made for large variations in return perigee distance which result from injection or midcourse-correction errors and still insure earth impact and, second, small changes in injection velocity and time can be made so that the flight time to the moon and the total mission flight time can be varied without a great deal of change in earth perigee distance. This is of importance in the design of a trajectory when it is necessary to have the vehicle in visual contact with a particular point on the earth's surface when the vehicle is in the vicinity of the moon, and also in establishing the position of vehicle-earth impact.

Utilization of such a trajectory (return perigee near the center of the earth) implies that the vehicle is falling nearly straight toward the earth when it leaves the moon's influence. The vehicle reentry angle will be approximately 90° and the reentry decelerations and heating rates will necessarily be high. The latitude of the reentry point will be approximately equal to the declination of the moon at the time of periselenian passage. Although this defines the range of earth latitudes available for reentry, it should be noted that the most favorable declination of the moon, or latitude of the reentry point, for a photographic experiment may be dictated by the position of the earth's radiation belts. In order to avoid these regions it is necessary to design the trajectory so that periselenian passage occurs near maximum positive or maximum negative declination of the moon.

In designing the actual trajectory, the factors mentioned must be integrated with the mission objectives. The photographic mission presented in this paper is considered for the period from the middle of 1962 to the end of 1963. Third-quarter lighting conditions were chosen as best suited for the photographic purposes of the mission, and reentry and recovery of the film package was scheduled to take place in the Pacific Ocean near Hawaii. This reentry location implies that the moon is at or near maximum positive declination at the time of periselenian passage. The simultaneity of these two conditions, that is, third-quarter lighting and maximum or near maximum positive declination, occurs in a 2- to 3-month period in the latter part of 1962 and similarly in the latter part of 1963. The trajectory presented in this paper is designed for the 1963 occurrences. It should be noted, however, that a similar trajectory may be designed with a target date in the early part of 1963. In this case, third-quarter lighting conditions would occur for maximum negative declination of the moon and the reentry point could most advantageously be located in the Indian Ocean near Australia.

The characteristics of the design or nominal trajectory are listed in table I. To obtain these results, use was made of a precision four-body, three-dimensional trajectory program (Encke method) utilizing actual calendar dates and body positions as obtained from unpublished U.S. Naval Observatory data.

The earth trace of the trajectory during the initial and final 24-hour periods of the mission is shown in figure 2. The launch phase is compatible with the capability of the Atlas-Agena B booster vehicle. The launch takes place at Cape Canaveral and injection occurs over South Africa after a specified coast period in a 115 statute-mile orbit. The entire injection burning phase will be visible from and under the control of the South African station of the Deep Space Network. The earth trace for the period from 68 to 72 hours is also shown. This is the time period during which photographs of the moon will be made and vidicon information transmitted to the earth. It is a requirement of the trajectory that this portion of the mission be visible from the Goldstone station. Since the trajectory has a perigee location which is not at the exact center of the earth, the trace of the trajectory has a turn-around point prior to reentry. In this case, the earth trace of the vehicle turns back on itself.

The spatial trace of the trajectory is shown in figure 3. The origin of the inertial coordinate system used in this figure is at the earth's center, and the system is oriented so that the X-axis is directed toward the moon at the time of injection, the Y-axis is perpendicular to the X-axis and is in the earth-moon plane, and the Z-axis is directed so as to form a right-hand system. A plot of the trajectory in the vicinity of the moon is shown in figure 4.

An error analysis has been made to determine the dispersion of the nominal earth-impact point due to uncertainties in vehicle tracking, the magnitude and direction of the midcourse correction, the earth-moon mass ratio, and the value of the geocentric gravitational constant. The results of this analysis are summarized in table II. Based on results obtained from the Jet Propulsion Laboratory, it was assumed that the vehicle midcourse correction capability (this correction is to be applied after the vehicle is acquired by the Goldstone station) was such that the miss distance at periselenian passage would be within a 30-mile-radius circle centered at the nominal periselenian distance and situated in a plane perpendicular to the trajectory plane. In order to determine the dispersion of the earth-impact point due to off-nominal trajectories lying within this miss circle, the injection conditions were varied from the nominal, and the earth-impact dispersion area corresponding to the 30-mile-radius lunar miss circle was determined. This dispersion area was elliptical in shape, with maximum variations of approximately 423 statute miles east and west and 282 statute miles north and south.

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Variations in the earth-moon mass ratio of ±0.1 (from the nominal value of 81.45) resulted in an east-west dispersion of the earth-impact point of approximately 85 miles. The dispersion due to an assumed uncertainty in the geocentric gravitational constant of 1 part in 90,000 was approximately 590 miles east and west and 138 miles north and south from the nominal point. These results must be combined statistically with the injection and midcourse-correction-error results to determine the total dispersion area. If the conservative approach of using the root sum square of the individual dispersions is adopted, the resulting dispersion of the impact point will be 730 miles east and west and 314 miles north and south.

This dispersion area is shown on the earth trace of the trajectory (fig. 2). It should be noted, however, that the midcourse correction maneuver will tend to reduce the dispersion due to the uncertainty in the gravitational constant and therefore the results given above are considered to be conservative.

Attitude Control of Photographic Axis and Lunar-Surface Coverage

Several possibilities have been considered for orientation of the camera axis for photographing the lunar surface. These possibilities include: (1) relocation of the midcourse-correction propulsion system and location of the photographic experiment in the propulsion-system position with an attitude control system on the photographic system, (2) a fixed location for the photographic system with the spacecraft attitude control system used to obtain the desired orientation (similar to the maneuver required prior to the midcourse velocity correction), and (3) a scheme which retains the present Ranger (RA-3) configuration with an orientation system on the photographic package, to be described subsequently. All these possibilities are feasible and could be considered in more detail before arriving at a final design.

In the interest of expediency, it is desirable to retain as much of the present Ranger configuration as possible, including the present location of the midcourse-correction propulsion system. It is also desirable to maintain the spacecraft in its cruise orientation during the photographing process, by means of the presently available, very accurate, spacecraft attitude control system. In the cruise orientation, the attitude control system maintains the spacecraft longitudinal axis pointing at the sun, the yaw axis in the earth-vehicle-sun plane with the high-gain antenna pointing at the earth, and the pitch axis perpendicular to this plane.

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Elimination of the landing capsule provides ample payload and volume within the present Ranger configuration for the photographic experiment. A scheme for orientation of the photographic axis which appears to warrant further consideration is as follows:

While retaining the spacecraft in its cruise attitude, rotate the photographic-experiment apparatus, including the reentry package, about an axis parallel to the pitch axis through a specified angle. This operation would be initiated upon command, 1 to 2 hours prior to the time when the vehicle is in the correct trajectory position for photographing the lunar surface. The photographic axis lies in the earth-vehicle-sun plane and is oriented in a particular inertial direction, and the course of the design trajectory carries the line of sight across the surface of the moon for photographing with the two cameras. The situation is illustrated in figure 4. Upon completing the first series of photographs, two additional rotations are made for obtaining two more passes of the line of sight across the surface, resulting in photographs from three different inertial attitudes for detailed analysis of surface features. The photography is completed before the vehicle loses line-of-sight contact with the earth. A drawing of the photographic package and reentry configuration, illustrating the orientation scheme, is shown in figure 1.

The projection of the trajectory on the earth-moon plane in the vicinity of the moon is shown in figure 4 which illustrates the coverage of the lunar surface obtained by the photographic axis in the three inertial positions. The numbers on the curve refer to time along the trajectory in hours from injection and distance from the center of the moon in miles. The first position allows photography from a distance of 13,000 to 9,000 miles from the center of the moon for a duration of $1\frac{1}{2}$ hours. Rotation about the actuator axis through an additional 18°

to the second position allows photography from about 8,500 miles to 5,700 miles from the center of the moon for about 1 hour, and further rotation through about 32° to the third position results in photographs from 5,000 miles to 4,000 miles for about 3/4 hour. The photographic experiment would be performed during a total time of about 4 hours.

As a consequence of restraining the plane of rotation of the actuator arm to coincide with the earth-vehicle-sun plane, an important consideration is where this plane, and consequently the photographic axis, intersects the surface of the moon. For the nominal trajectory in figure 4, the lines of intersection of the photographic axis and the surface of the moon, for the three inertial attitudes, are shown in figure 5, which also illustrates the relative fields of view at the indicated times. The traces are in the northern hemisphere of the moon, commencing in the southern portion of Mare Imbrium and extending through the northern portion of Oceanus Procellarum on the visible side of the moon, and are approximately parallel to the lunar equator. The series of high-resolution photographs will cover strips of the lunar surface centered about these lines, while the wide-angle photographs will cover the complete surface.

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The traces shown in figure 5 are considered to be acceptable, but do not intersect any major lunar craters, with the possible exception of Archimedes, which would be visible from a highly oblique angle on the western limb. The traces could be shifted toward the equator, with the possibility of viewing the craters Eratosthenes, Copernicus, or Kepler with the high-resolution camera, by the simple expedient of mounting the camera axis at a small angle to the plane of rotation of the actuator arm. This angular offset can be obtained from predetermined trajectory characteristics and set to the desired value during prelaunch assembly of the payload package. Such a procedure would also compensate for differences in trajectory characteristics for nominal trajectories designed for different launch dates from the one shown here. In the present case an angle of 3° between the photographic axis and plane of rotation of the actuator arm (i.e., the earth-vehicle-sun plane) would cause the lines of intersection of the photographic axis and the lunar surface to be situated near and approximately parallel to the lunar equator. Overlapping fields of view of the high-resolution camera for the three inertial attitudes are obtained with an angular offset of $1/2^{\circ}$. The surface coverage of the high-resolution camera and the fields of view are illustrated in figure 6.

The command for initiation of photographing can be given either at preset times, from knowledge of the trajectory and known values of the rotation angles of the photographic package, or by a sensor pointing along the photographic axis which gives a signal when the moon is in view.

It would be desirable to include the vidicon photographic system, already developed for Ranger (RA-3), on the film photographic package, so that relatively low-resolution vidicon pictures of the lunar surface would be transmitted to earth during the flight as an auxiliary experiment.

Photographic System Design Considerations

Slightly over one-half of the moon's surface is visible from the earth and has been explored extensively by telescopes, both visually and photographically. The best photographic methods will resolve lunar surface features which are about 3,000 feet in diameter; visual resolution depends on the observer but under best seeing is generally about 1,200 feet. In 1959 the Union of Soviet Socialist Republics (USSR) obtained television pictures of part of the back side of the moon by photographing it at an age of 11 days and transmitting the pictures back to the earth. A considerable loss in resolution results when the data require transmitting in this manner as indicated by the released photographs which have a resolution of the order of tens of miles (ref. 3). At the present time there are available pictures of medium resolution of 59 percent of the moon's surface and pictures of poor resolution of 25 percent, with 16 percent of the moon's surface unmapped. This unmapped region, along with the entire eastern limb, can be photographed at third quarter.

The detailed design of an optical system consists of a series of compromises and optimizations. In the interest of expediency one of the primary design criteria will be the utilization of as much "off the shelf" equipment as possible and the discussion of the photographic considerations will be started in that direction. The purpose of this section is not to present a final design, but rather to present the results of a feasibility study in the form of an optical-parameter analysis. The ground rules for the analysis are that the optical system must fit into the Ranger vehicle and must weigh less than about 100 pounds.

For close-in photography of the lunar surface, a compromise must be made between angular field and resolution. Consideration of available optical and photographic equipment and the allowable weight for the optical system leads to the conclusion that no single system would photograph the entire disk of the moon with high resolution. Thus, at least two systems should be used, one having a wide field and the other having a narrow field with high resolution. These conditions lead to the choice of a refractor system for the wide-angle camera and a reflector system for the high-resolution camera.

For the wide-angle system (the moon subtends a maximum angle of 30° for the trajectory discussed in other sections) an anastigmatic refractor is recommended since it has the advantage of reduced coma and astigmatism for off-axis objects. However, on-axis resolution must be compromised since refractors seldom approach Dawes diffraction limit. The high-resolution system will have a very narrow field so that astigmatism and coma are of less importance and use of a corrected parabolic reflector is recommended.

The main component of the refracting system would be a standard 80-mm focal length f/4 wide-angle lens. This type of lens has been produced with a photographic resolution of 100 lines/mm over a field of 30°, and would require 70-mm film. Assuming a resolution of 100 lines/mm, the resolved distance on the surface of the moon can be calculated as a function of the altitude above the moon or, equivalently, the flight time for the nominal trajectory. The results of these calculations are presented in the upper curve of figure 7. Notice that most of the pictures will have better resolution than the present lunar maps.

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The design consideration of the reflecting system must include a number of factors. First, the optical axis is translating relative to the surface of the moon so that during the time of exposure a smearing of the image occurs. The smear distance is the velocity of the vehicle normal to the optical axis times the exposure time. The exposure time to produce an exposure of E (film density = 1) is a function of the optical system and the film and may be expressed as

$$t = \frac{4}{\pi} \frac{f_e^2 E}{B\tau}$$

where

t	required exposure time
f_e	effective focal ratio
E	exposure required (meter-candle)(sec)
B	surface brightness of the moon, 0.25×10^4 candles/m^2
τ	transmission coefficient of the optical system, taken as 0.90 for this study.

A complete list of symbols used in this paper is given in the appendix.

The smear[1] in feet for a velocity of V ft/sec is

$$S_m = \frac{4}{\pi} \frac{f_e^{2}EV}{B\tau} \tag{1}$$

A second consideration is the resolving power of the film. For a film with a resolving power of N lines/mm, the distance resolved on the lunar surface, assuming perfect optics, is

$$L_m = \frac{528h}{Nf_eD} \tag{2}$$

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where h is the height above the lunar surface in miles and D is the objective diameter in cm. A third consideration is the image degradation due to diffraction at the edge of the objective. Diffraction limits the surface resolution to

$$R_m = \frac{0.3216h}{D} \tag{3}$$

Although no specific method has been derived for combining resolutions, an approximate method is to take the square root of the sum of the squares. The final resolution is therefore

$$R = \sqrt{{L_m}^2 + {S_m}^2 + {R_m}^2}$$

Substitution from equations (1), (2), and (3) yields

$$R = \sqrt{\left(\frac{528h}{Nf_eD}\right)^2 + \left(\frac{4f_e^{2}EV}{\pi B\tau}\right)^2 + \left(\frac{0.3216h}{D}\right)^2} \tag{4}$$

The purpose of the parametric study is to minimize R within the weight and size limitations. The most obvious method of minimizing two of the terms of equation (4) is to make D very large. However, the weight of a reflecting mirror increases approximately as the cube of the mirror diameter. For a 12-inch mirror the weight is between 20 and 25 pounds and for an 18-inch mirror the weight is about 70 pounds. Also, incorporating the larger mirrors into the Ranger spacecraft will be more difficult. For this study the objective diameter is taken to be 12 inches.

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[1]The smear due to the angular rates imposed by the attitude control system has been calculated and found to be an order of magnitude less than the values obtained from equation (1).

Certain of the parameters in equation (4) are coupled; for example, the velocity and altitude are connected by an energy integral so that, as h increases, V decreases. For specified film and optical parameters, there exists an optimum altitude for photographing the moon. For a typical low-speed film with a 12-inch f/11 objective and typical circumlunar energies, a very flat minimum occurs at 1,600 miles above the moon's surface. Over an altitude range from 0 to 4,000 miles, the resolved distance changes by only 10 percent. Therefore, altitude optimization does not present any large advantages.

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A second coupling of the terms in equation (4) concerns the film characteristics. Although an explicit relationship between E and N would include such intangible quantities as developing techniques, in general, N is a monotonic increasing function of E. Thus for each optical system there is some film which gives the best resolution. In addition, it is apparent from equation (4) that there exists an optimum focal ratio for each film. Three typical films were chosen to examine the effect of film type and focal ratio. The results of the study are summarized in figure 8 where lunar-surface resolution is plotted against focal ratio with film type as a parameter. As would be expected, at high focal ratios, low-resolution—high-speed film is required and at low focal ratios, high-resolution—low-speed film is required. The minimum resolution for each film type seems to be decreasing slowly as film speed increases. The system with the high focal ratio, however, is more difficult to fold into the vehicle. In fact, for maximum utilization of the optical-system resolving power, Cassegrainian focus will probably be required. Utilizing very-high-resolution film and allowing some reduction from optimum resolution may relax the requirement of folded optics. However, this compromise can only be decided after a detailed analysis. In either case, surface resolution will be about 100 feet, as shown on the lower curve on figure 7 and therefore considerably better than the 3,000-foot resolution of the lunar surface available from the earth.

Calculating the field of view requires a specific set of optical characteristics. For example, the f/11 system with the 12-inch mirror has an angular field of about 1° on 70-mm film. This is equivalent to a region on the lunar surface of 100 miles square when the vehicle is 6,000 miles above the surface. To obtain overlapping high-resolution photographs from the positions closer to the moon, the photographs must be taken at the rate of about one per minute. If this constant rate is used, about 240 pictures will be taken, and about 60 feet of 70-mm film will be required for the high-resolution photographs.

Another important consideration in a high-performance optical system is that of maintaining the optical resolution under changing temperature conditions. Reference 4 presents a detailed analysis of the problem and possible solutions. Spitzer divides the problem into two parts. First, the effect of local thermal inhomogeneity on mirror figure and, second, the effect of thermal expansion on the relative positions of the optical elements. If low-conductivity mirror supports are utilized, the first problem is not serious since the time required to reduce internal temperature differences a factor of 1/e by conduction is very much less than the corresponding time for radiation cooling to produce the same temperature reduction, infinite conductivity being assumed.

The effect of thermal expansion on the relative positions of the optical elements may require some design studies. However, since the orientation of the optical system relative to the sun does not change during the cruise phase, the optical system will be in equilibrium and nearly isothermal. Careful design of the package to nearly maintain these conditions during the photographic phase in conjunction with the techniques outlined by Spitzer should alleviate the thermal-expansion problem. These techniques include (1) making the entire optical system of one material so that the system contracts and expands proportionally, (2) making the system of materials with low thermal expansion coefficients, and (3) using materials of different thermal expansion coefficients but assembling components in such a manner as to compensate for temperature changes.

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Radiation Protection of Film

Any experiment which involves the transport of photographic film beyond the protective cover of the earth's atmosphere must consider means of protecting the film from radiation damage due to passage through the Van Allen belts and from other sources of radiation in space. The allowable radiation dose for film is much less than that for men. Reference 5 indicates that a dose of 10 roentgens would produce intolerable blacking of typical amateur and commercial films, a dose of 4 roentgens would produce a density of 1, and a dose of 1 roentgen would produce a density of about 0.4. This latter density occurs at the "foot" of the exposure-density curve and this may be taken as the maximum allowable density before image deterioration begins. Thus doses of 1 roentgen or less will be acceptable.

There are a number of space radiations which must be considered

(1) Protons in the inner Van Allen belt

(2) Electrons in both inner and outer Van Allen belts

(3) Solar-flare particles

(4) Primary cosmic rays

(5) Secondary radiation from the interaction of the preceding with the structure.

The effects of the secondary radiation produced by the protons, solar-flare particles, and cosmic rays have not been investigated in any detail. Some preliminary calculations indicate that the dose due to the secondary radiation produced by the protons can be as great as 50 percent of the primary dose and for cosmic rays can be as great as 100 percent of the primary dose. More detailed calculations will have to be performed in order to evaluate the precise doses.

Reference 6 indicates that the biological dose from solar flares can range from a negligible amount to about 25 roentgen behind 25 g/sq cm of water for low- and high-energy flares, respectively. The latter dose would produce almost complete blackening of the film. In addition, a shielding of 25 g/sq cm of H_2O cannot be tolerated from a weight standpoint, for the shielding of a film box with inside dimensions of 4 by 4 by 8 inches would weigh about 600 pounds. This leads to the conclusion that the occurrence of a high-energy solar flare would ruin the film. Fortunately, a solar flare minimum occurs in the 1963 and 1964 time period, so that the number of major flares expected during the proposed launch periods is nearly a minimum. Extrapolating the curve of reference 6 for flares per month to 1962 and 1963 period indicates an expectation rate of much less than 1 flare per month. For a 1-week flight the probability of having a major flare is therefore small.

The proton dose in the center of the inner Van Allen belt has been calculated in reference 6. With a particle flux of 20,000 protons/(sq cm)(sec) with energies greater than 40 Mev, the dose rates behind 2 g/sq cm and 25 g/sq cm of water are 12 r/hr and 2.7 r/hr, respectively. Again the 25 g/sq cm must be eliminated from consideration from a weight standpoint. Since 2 g/sq cm of water shielding the film box only weighs 5 pounds, it is of interest to determine what dose the film would be exposed to with this amount of shielding.

To calculate this dose requires a knowledge of the spatial dependence of the proton flux and of the vehicle trajectory. Figure 9 illustrates the proton counting rate, normalized to unity at the center of the belt, as a function of the geomagnetic latitude and the geocentric radius. The ascending and descending portions of the circumlunar trajectory discussed in a previous section are also illustrated. The trajectory has been designed to miss the belt, consistent with the constraints imposed by communication and kinematic requirements. The integrated dose for both passes through the inner belt, including 50 percent for secondary radiation, is about 1/2 roentgen. The reduction in dose afforded by trajectory shaping is illustrated by comparing the 1/2-roentgen dose with a dose of 4 roentgens obtained for trajectories in which the vehicles leave and return through the center of the belt.

If required, further reduction in the dose can be obtained by designing the trajectory so that the vehicle enters at a greater geomagnetic latitude than the nominal trajectory presented here. The magnetic equator dips south over South America and north over the Indian Ocean. Thus, from a radiation standpoint, the two most favorable landing regions are in the eastern Caribbean and almost anywhere in the Indian Ocean or Australia.

The electrons in both belts are less penetrating than the protons in the inner belt. These electrons will be stopped by the structural mass, and the inside of the vehicle will be irradiated by the resulting bremsstrahlung. Utilizing a material having a low atomic number for the vehicle skin reduces the intensity of the bremsstrahlung. However, even with materials having medium atomic numbers, the radiation can be attenuated to a negligible intensity by 2 or 3 g/sq cm of lead. Thus the electrons do not present any great radiation problems.

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The radiation dose rate due to primary cosmic rays is about 1.6×10^{-3} r/hr, giving a total dose for a 160-hour trip without shielding of only 0.25 roentgen. Considering secondary radiation the total will probably be less than 0.5 roentgen. Shielding against the low-atomic-weight components of the cosmic rays is quite impractical; however, due to the nominal total dose, shielding is not mandatory.

To summarize, the approximate doses within an aluminum film-protection container with 1.1-centimeter-thick walls (equivalent shielding of 2 g/sq cm of water) and 3 millimeters of lead shielding inside the aluminum are shown in table III. The weight of such a container with inside dimensions of 4 by 4 by 8 inches is approximately 12 pounds. It is seen that this amount of shielding reduces the radiation dosage to the acceptable level of about 1 roentgen and protects the film against all space radiations except a major solar flare.

REENTRY, HEATING, AND RECOVERY CONSIDERATIONS

The circumlunar trajectory design incorporating vertical entry into the earth's atmosphere on return to earth leads to reentry conditions characterized by large decelerations and high heating rates. Both conditions, however, are tolerable for the reentry of a recoverable instrumented vehicle.

A number of reentry trajectory and heating calculations have been made for reentry configurations which could be utilized for the photographic mission. Although not necessarily the final choice, a spherical reentry configuration has the advantage of requiring no attitude control after separation from the vehicle, prior to the reentry heating period. Typical trajectory properties for a spherical reentry configuration with the reentry parameter W/C_DA of 50 lb/sq ft and for vertical entry into the atmosphere with velocity of 36,000 ft/sec at an altitude of 76 miles are shown in figure 10. Time is measured from time of passage through an altitude of 76 miles. Peak deceleration is 354g. The vehicle has decelerated to sufficiently low velocity (<760 ft/sec) for parachute deployment at altitudes less than about 76 miles.

Basic data for estimation of heat-shield weight are shown in figure 11 as a plot of heating rate $\dot{Q}$ as a function of time for the convective and radiative components, calculated for a standard 1-foot nose radius and $\frac{W}{C_DA} = 50$. These data are used to calculate the total convective and radiative heat inputs for the $1\frac{1}{2}$-foot-diameter reentry sphere and result in approximately 15 pounds of ablative heat material required for an ablative heat shield (phenolic nylon) with effective heat capacity of 5,000 Btu/lb. Approximately an equal amount of material will be required to maintain the temperature within the reentry capsule at less than about 100° F throughout the reentry; the result is a total heating-protection weight of 30 to 35 pounds.

The recovery of the reentry capsule containing the undeveloped film is somewhat simplified in that the actual impact point, within the overall dispersion area, should be predictable to a high degree of accuracy. The Ranger spacecraft will be tracked throughout the circumlunar trajectory and, with the single midcourse correction having been made less than 1 day after injection, more than 6 days of continuous tracking and smoothing time will be available. It is estimated that the impact location will be predictable within a few tens of miles 2 days before reentry, and within a few miles several hours before reentry. In contrast to recovery from an earth orbit, there is no requirement for a deorbiting retro-rocket and attendant attitude and timing control, and with the proposed nearly vertical reentry trajectory, there will be very little dispersion due to atmospheric effects.

It is of some interest to note that a vehicle describing a circumlunar trajectory such as the one presented here, in which the vehicle passes near the moon at the third-quarter lighting condition, and returns to the earth nearly normal to the atmosphere, enters the atmosphere at the time of local dawn, thus affording a maximum number of daylight hours for the recovery operation.

The recovery aids, including a radar-reflective buoyant parachute, a radio beacon operable during parachute descent and while the inherently buoyant capsule is floating on the water, visual aids, and sequencing devices are all presently available and have been proven in previous recovery operations. Similarly, the recovery operation, which might include a consideration of air snatch, should follow previously established procedures but, in consideration of factors mentioned above, should be of less complexity than recoveries from earth-orbital missions.

PRELIMINARY DESIGN OF PHOTOGRAPHIC EXPERIMENT

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An overall view of the photographic-experiment design and its integration into the Ranger spacecraft is shown in figure 1. The mounting and actuator for positioning of the photographic package are the only additions required to the standard Ranger (RA-3) bus. This actuator is mounted on the opposite side of the base of the bus from the mounting and actuator for the high-gain antenna, and the photographic package is deployed in the same plane as the high-gain antenna. The supporting structure for the omnidirectional antenna, which is fixed for this mission, is attached to the bus.

Figure 12 is a preliminary drawing of the experimental package design, showing the high-resolution camera, wide-angle camera, film transport and shutter mechanisms, the reentry capsule, and associated components. Consultation with respect to the design of the high-resolution camera has indicated that such a design, including consideration of the thermal environment, is well within the present state of the art.

The preliminary weight estimate for the experimental apparatus is given in table IV. The estimated weight of 239 pounds is well within the allowable weight of 300 pounds, and indicates the possibility of inclusion of additional experiments on the circumlunar mission.

ANTICIPATED SCIENTIFIC RESULTS

Photographs of the lunar surface obtained from a circumlunar photographic experiment would provide basic data for a number of scientific investigations. With surface coverage of the entire eastern limb down to a resolution of 2,000 feet, obtained with a wide-angle refractor, and with coverage of a wide area north of the equatorial region (approximately 0.5×10^6 square miles) with resolution down to approximately 80 feet, obtained with a high-resolution reflector, a considerable amount of new information would be available pertaining to the lunar-surface features and formations. These photographs would cover that portion of the back side of the moon which has not yet been photographed and would cover one-half of the back side with considerably better resolution than the available pictures.

Since the photographs would be taken with the photographic axis in three different inertial attitudes, data would be available for determination of altitudes and slopes of surface features at the two scales of resolution.

Photometric studies, following the approach of Bennett (ref. 7) and other investigators and more recently by Van Diggelen (reported by Struve in ref. 8), should provide new information pertaining to the texture of the lunar surface. Of particular interest in this connection is whether the curves of surface brightness as a function of phase angle (here defined as the angle between the direction of the sun and the photographic line of sight) are of the same nature for features resolved with the high-resolution camera as for features resolved with present earth-based photographs.

The wide-angle pictures taken about 70 hours after injection would have a resolution of about 5,000 feet and give a view of the entire eastern limb of the moon along with a profile of the figure of the moon in the meridian plane containing the earth. These pictures would help to determine the size of the lunar bulge toward the earth. As indicated in reference 9, there is considerable uncertainty in the value of the ellipticity of the moon's figure. Dynamical considerations give a length of the semiaxis alined with the earth 3,140 feet greater than the average lunar radius normal to the line of sight. Measurements of apparent position of lunar-surface features against the star background at different librations give semiaxis elongations varying from 3,000 to 13,000 feet. The most accurate measurements yield 6,500 feet ± 2,200 feet. Therefore, a statistical analysis of the wide-angle photographs should be able to reduce the uncertainty in the size of the lunar bulge.

Link (ref. 10) has proposed that lunar occultations of radio sources give an opportunity to determine the electron gas density at the surface of the moon and thereby to obtain some measure of the lunar atmospheric density. The technique is to measure the difference between the times of radio and visual occultation of the source. Elsmore (ref. 11) gives the results of such an experiment which he performed in 1956 when the moon occulted the Crab nebula. The angle of refraction was measured to be 13.4 seconds ± 8.7 seconds, corresponding to an electron density of the order of 10^3/cu cm, the uncertainty probably arising from the finite size of the radio source.

A circumlunar mission gives an excellent opportunity to reperform Link's experiment with a considerable improvement in accuracy, if optical tracking of the vehicle is available, for the satellite will represent coincident radio and optical point sources and the times of occultation can be measured to a fraction of a second. Optical tracking might be afforded by placing a plane reflecting surface on the spacecraft and rotating it about the pitch axis so that the normal to the mirror bisects the sun-vehicle-observer angle. This rotation might be obtained by linking the mirror to the high-gain antenna. As indicated in reference 12, a mirror 13 inches in diameter is visible against the full moon when viewed through a 40-inch telescope. To obtain a high degree of accuracy, a transmitter with a lower frequency than the present Ranger communication system is required. Data in reference 10 indicate that any frequency less than 100 mc/sec would be suitable. To obtain refraction data at both the occultation and reappearance, the transmitter would have to operate for about 1 hour.

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The determination of accurate values of the masses of the earth and moon from an analysis of the actual vehicle trajectory is largely a function of the accuracy with which the vehicle can be tracked. To obtain an indication of the magnitude of the numbers involved, with the present trajectory an error of 1 part in 90,000 in the value of the geocentric gravitational constant, on the assumption of perfect injection, results in a vehicle position error of approximately ±15 miles after 23 hours of flight. The distance of the vehicle from the center of the earth is approximately 126,000 miles at this time. Since the midcourse correction will be made prior to this time, the tracking system must be able to resolve errors which are smaller than this value. In addition, the effect of errors in injection conditions must be separated from the effect of errors due to mass uncertainties.

An increase in the accuracy of the value of the lunar mass may be obtained by observation of the vehicle trajectory in the vicinity of the moon. In order to obtain an indication of the tracking accuracy which would be required, a variation of ±0.1 (from the nominal trajectory value of 81.45) was made in the earth-moon mass ratio. This variation had little or no effect on the trajectory prior to the vehicle's entry into the moon's sphere of influence. On the earth-return portion of the mission, however, this variation in the earth-moon mass ratio resulted in a vehicle position error of ±35 miles at a distance of 32,000 miles from the center of the moon. Although this variation in the nominal value of the earth-moon mass ratio covers the range of absolute values which have been obtained for this ratio (see ref. 13), the variation is about four times the expected uncertainty of any particular value. Therefore it appears that the position of the vehicle would have to be known to a value somewhat smaller than 8 miles if a more accurate value is to be obtained for the lunar mass.

CONCLUDING REMARKS

Analyses and design considerations generated in the course of a feasibility study indicate that an area of about 500,000 square miles of the lunar surface could be photographed with considerably better resolution than that obtained from earth-, balloon-, or earth-satellite-based instruments, by using a circumlunar trajectory and returning the photographic film to the earth. The study also indicates that trajectory optimization leads to reasonable dispersion areas on the surface of the earth on the completion of such a mission, with a consequent reasonable probability of effecting recovery of the data capsule after return to the earth. In addition to the primary objective of obtaining the lunar-surface photographs and vidicon pictures, which would provide basic data for analysis of lunar-surface features, formations and structure, a number of other scientific results would be obtained from such a mission.

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Langley Research Center,
National Aeronautics and Space Administration,
Langley Air Force Base, Va., January 15, 1962.

APPENDIX

SYMBOLS

In cases where distances are expressed in miles, the statute mile is intended. The following factors are included for use in converting English units to metric units: 1 statute mile = 1.609344 kilometers, 1 foot = 0.3048 meter, 1 inch = 2.54 centimeters.

B	surface brightness of moon, candles/m^2
D	objective diameter
E	exposure, (meter-candle)(sec)
f	focal ratio
f_e	effective focal ratio
h	height above lunar surface, statute miles
L_m	surface resolution limit due to film, ft
N	resolving power of film, lines/mm
$\dot{Q}$	heating rate, Btu/(sq ft)(sec)
$\dot{Q}_{c,max}$	convective heating rate, Btu/(sq ft)(sec)
R	resolution
R_m	surface resolution limit due to diffraction, ft
S_m	surface resolution limit due to smear, ft
T	time measured from injection, days
t	required exposure time
W/C_DA	ballistic parameter, lb/sq ft
V	velocity, ft/sec
X,Y,Z	inertial coordinate axes
x,y,z	distances along inertial coordinate axes
x_r,y_r,z_r	distances along rotating coordinate axes
ρ	density
τ	transmission coefficient of the optical system

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L-1880

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TABLE I.- TRAJECTORY CHARACTERISTICS

Launch conditions:	
Latitude of Cape Canaveral, deg N.	28.278
Longitude of Cape Canaveral, deg W.	80.574
Heading angle, deg	102.16
Time to injection, sec	2,097.01
Angular travel to injection, deg	127.689
Injection conditions:	
Time	October 5, 1963 $16^h50^m30.00^s$ u.t.
Latitude, deg S.	25.875
Longitude, deg E.	31.372
Heading angle, deg	106.896
Altitude, statute miles	115
Velocity, ft/sec	35,921.0
Elevation angle, deg	1.66
Periselenian conditions:	
Time from injection, hr	72.40
Distance of periselenian (from center of moon), statute miles	3,746
Reentry conditions at an altitude of 76 statute miles:	
Time from injection, hr	167.81
Latitude, deg N.	23.43
Longitude, deg W.	145.20
Reentry angle, deg	79.34

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TABLE II.- SUMMARY OF DISPERSION-AREA ANALYSIS

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Source	Dispersion of earth-impact point, statute miles	
	East and west	North and south
30-mile-radius miss circle (tracking and midcourse correction uncertainty)	423	282
Uncertainty in earth mass, $\frac{1}{90,000}$	590	138
Uncertainty in earth-moon mass ratio, $\frac{0.1}{81.45}$	85	Negligible
Root-sum square	730	314

TABLE III.- APPROXIMATE RADIATION DOSE INSIDE FILM CONTAINER

Source	Approximate dose, roentgen
Van Allen protons	0.5
Van Allen electrons	<.1
Cosmic rays + secondary radiation	<.5
Solar flare (Probability of occurrence during mission is small.)	>>1

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TABLE IV.- PRELIMINARY WEIGHT ESTIMATES

[All weights in pounds]

Reentry capsule:	
Film and radiation protection, take-up	20
Heat shielding and structure	45
Recovery aids (beacon, parachute, dye)	16
Total	81
Optical systems:	
Reflector (12-inch) system and structure	60
Refractor, shutters, drive, and structure	35
Radiation protection	13
Total	108
Supporting arm and actuator	35
Additional vehicle structure	15
Total weight	239

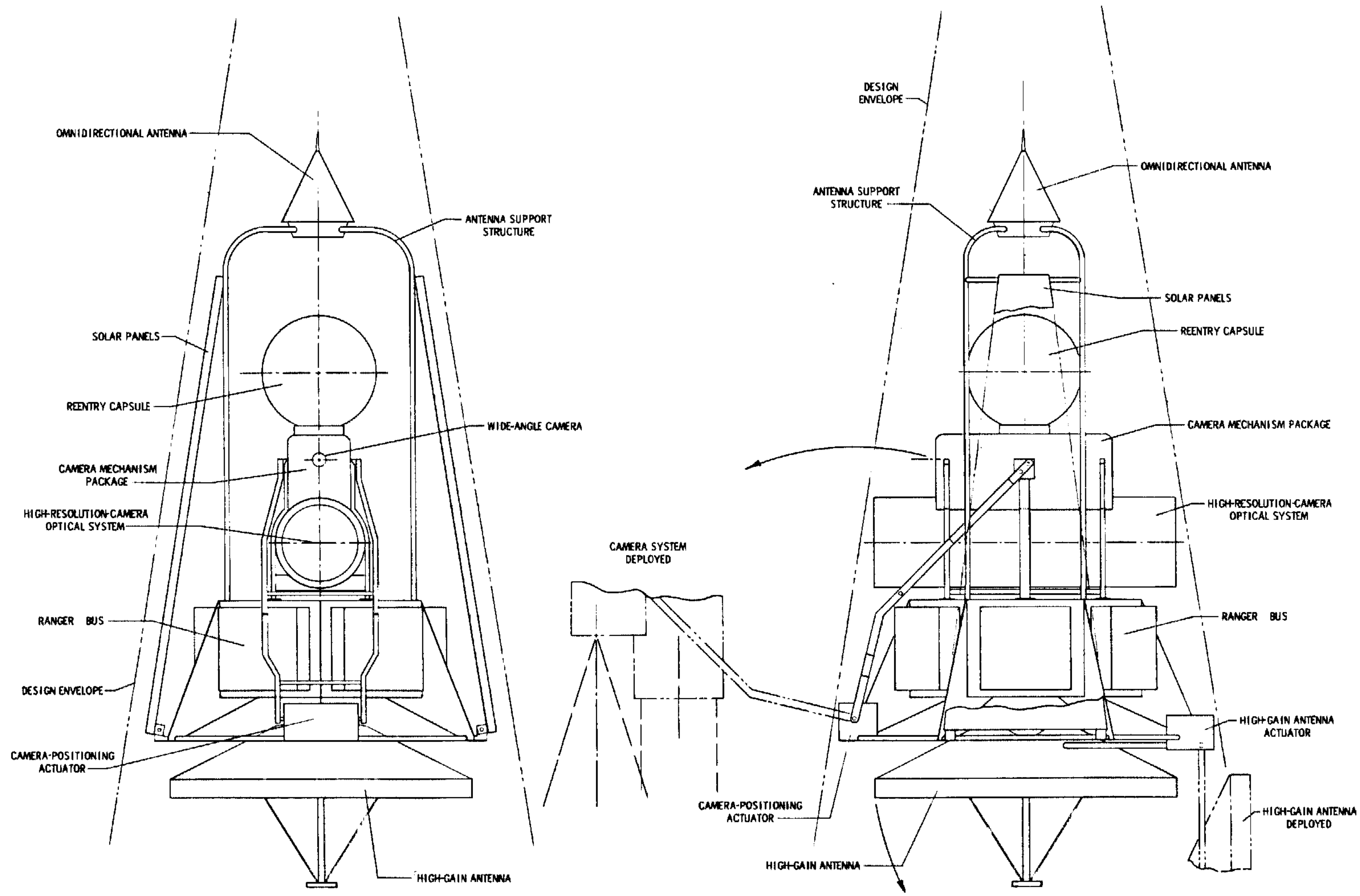

Figure 1.- Photographic experiment design.

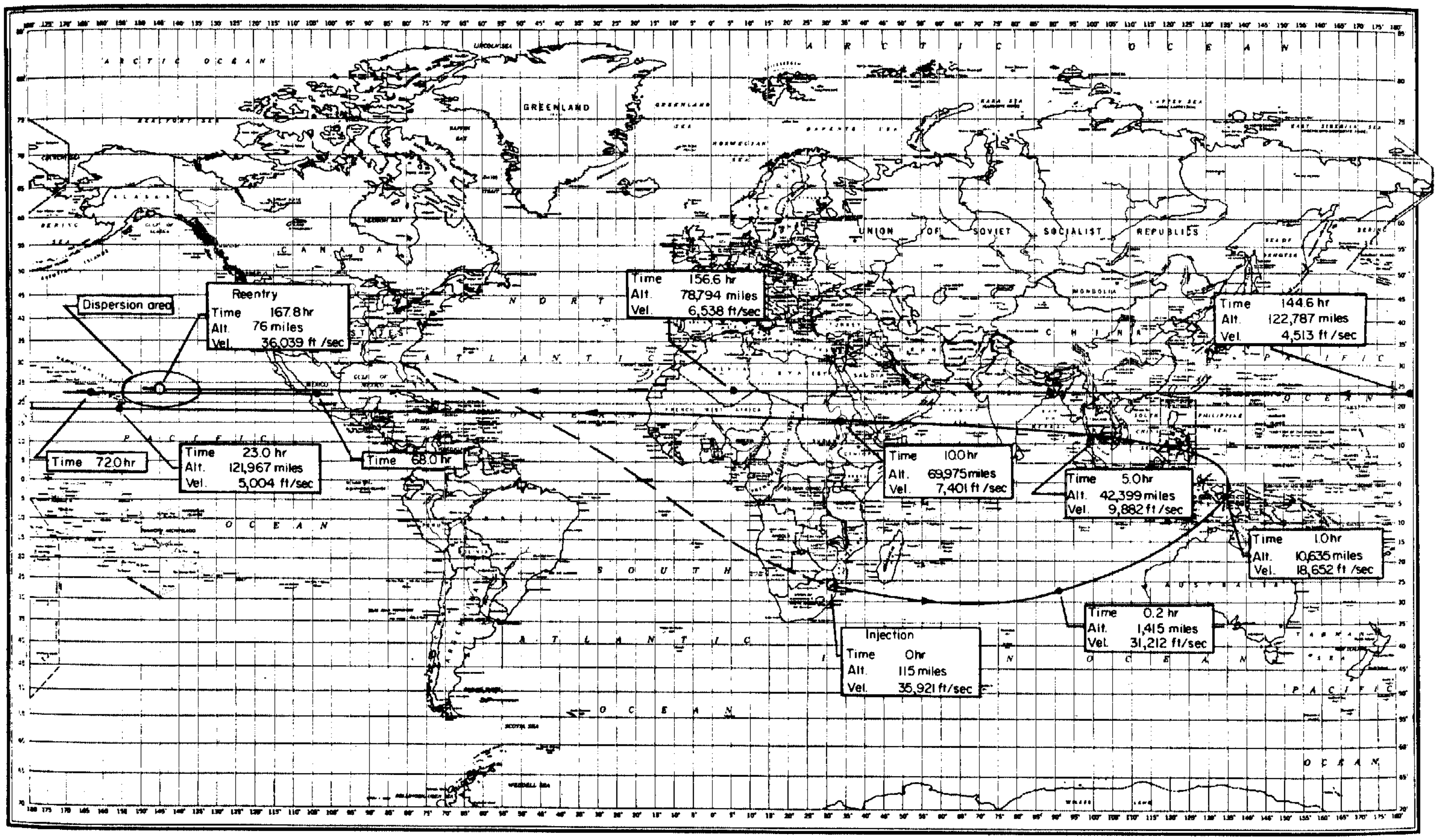

Figure 2.- Earth trace of nominal trajectory.

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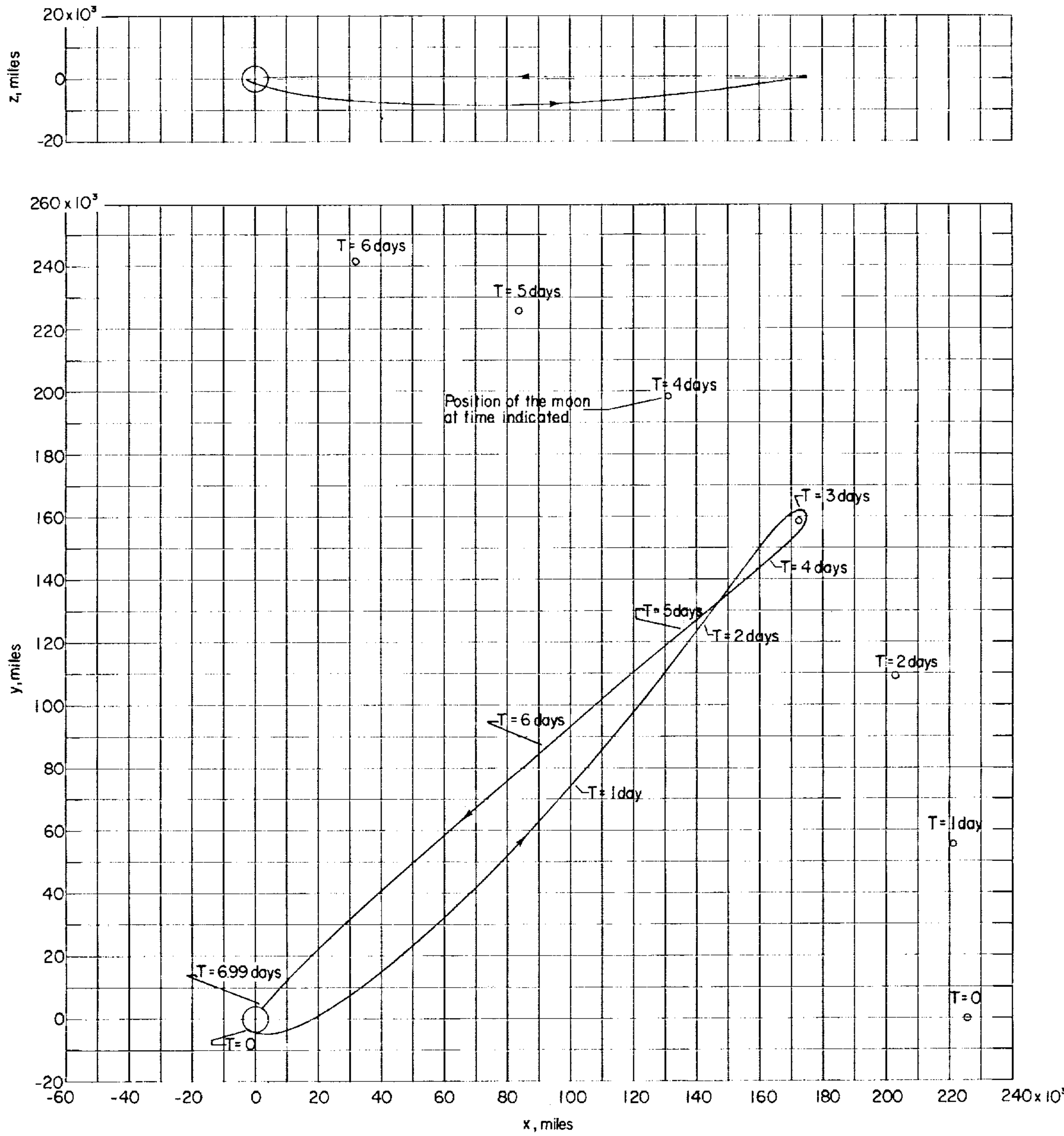

Figure 3.- Spatial trace of nominal trajectory. (Projections in earth-moon plane and in a plane perpendicular to earth-moon plane.)

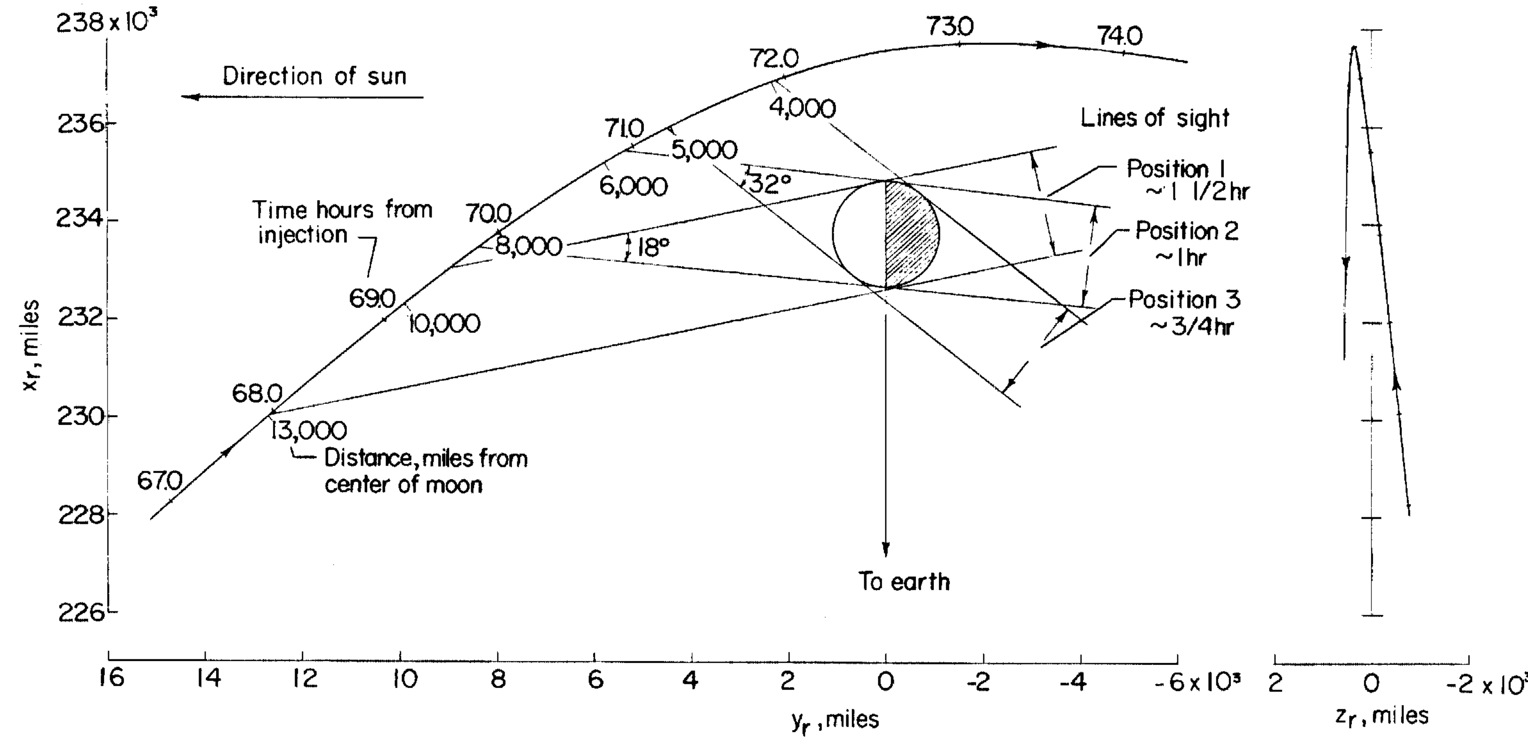

Figure 4.- Trajectory characteristics near the moon. (Projections in earth-moon plane and in a plane perpendicular to earth-moon plane.)

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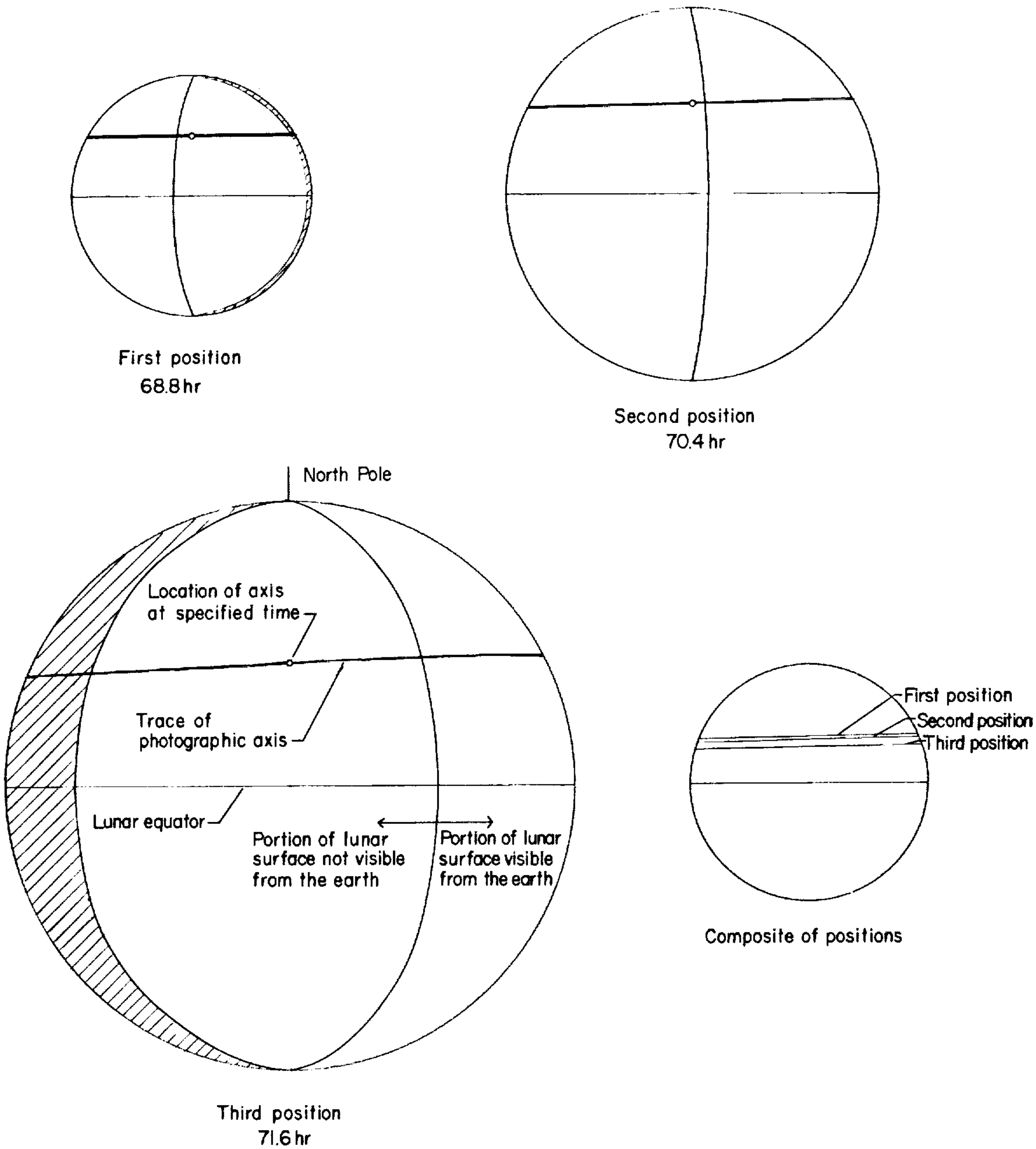

Figure 5.- Intersection of photographic axis with surface of moon.

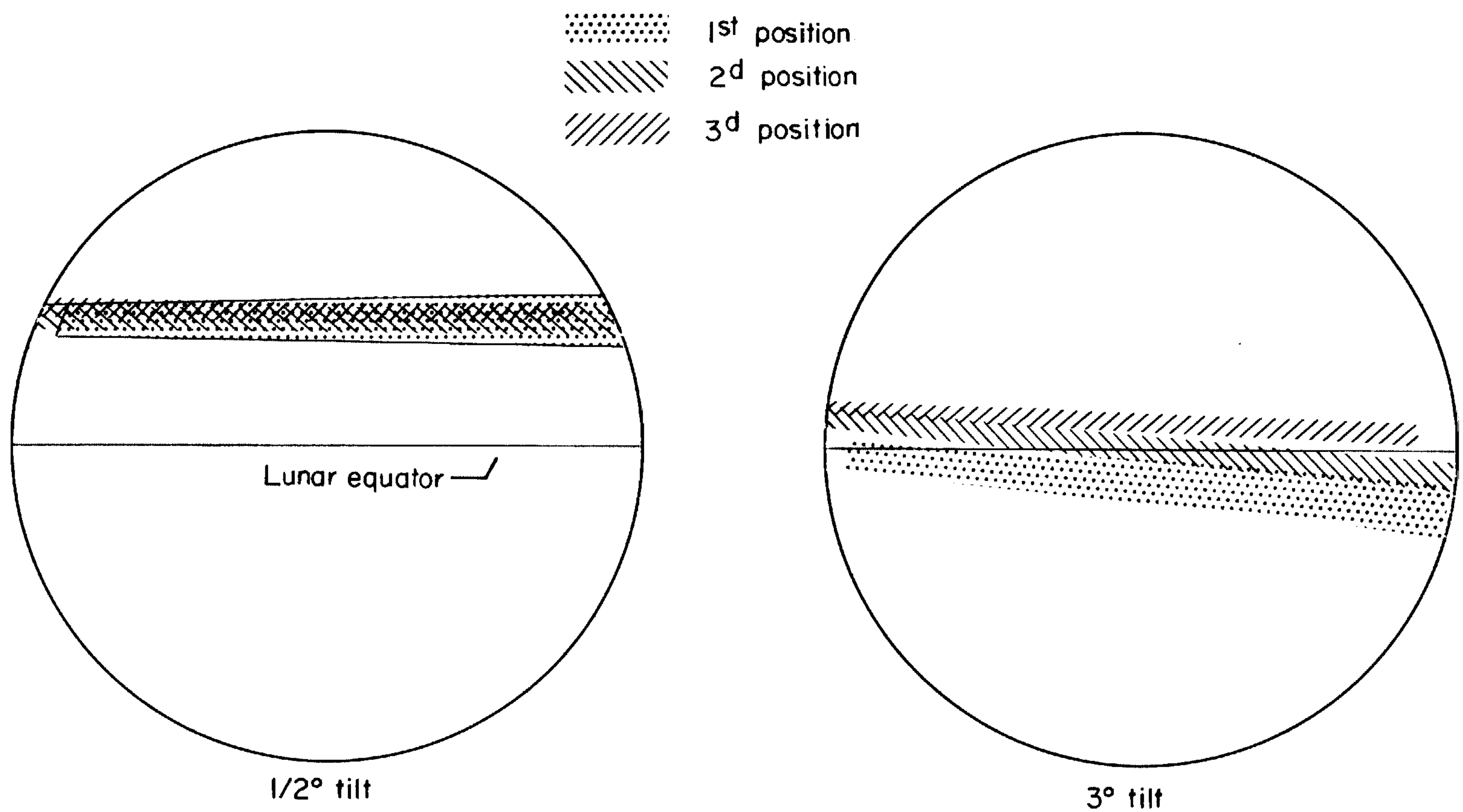

Figure 6.- Lunar-surface coverage with high-resolution camera for two orientations of photographic axis.

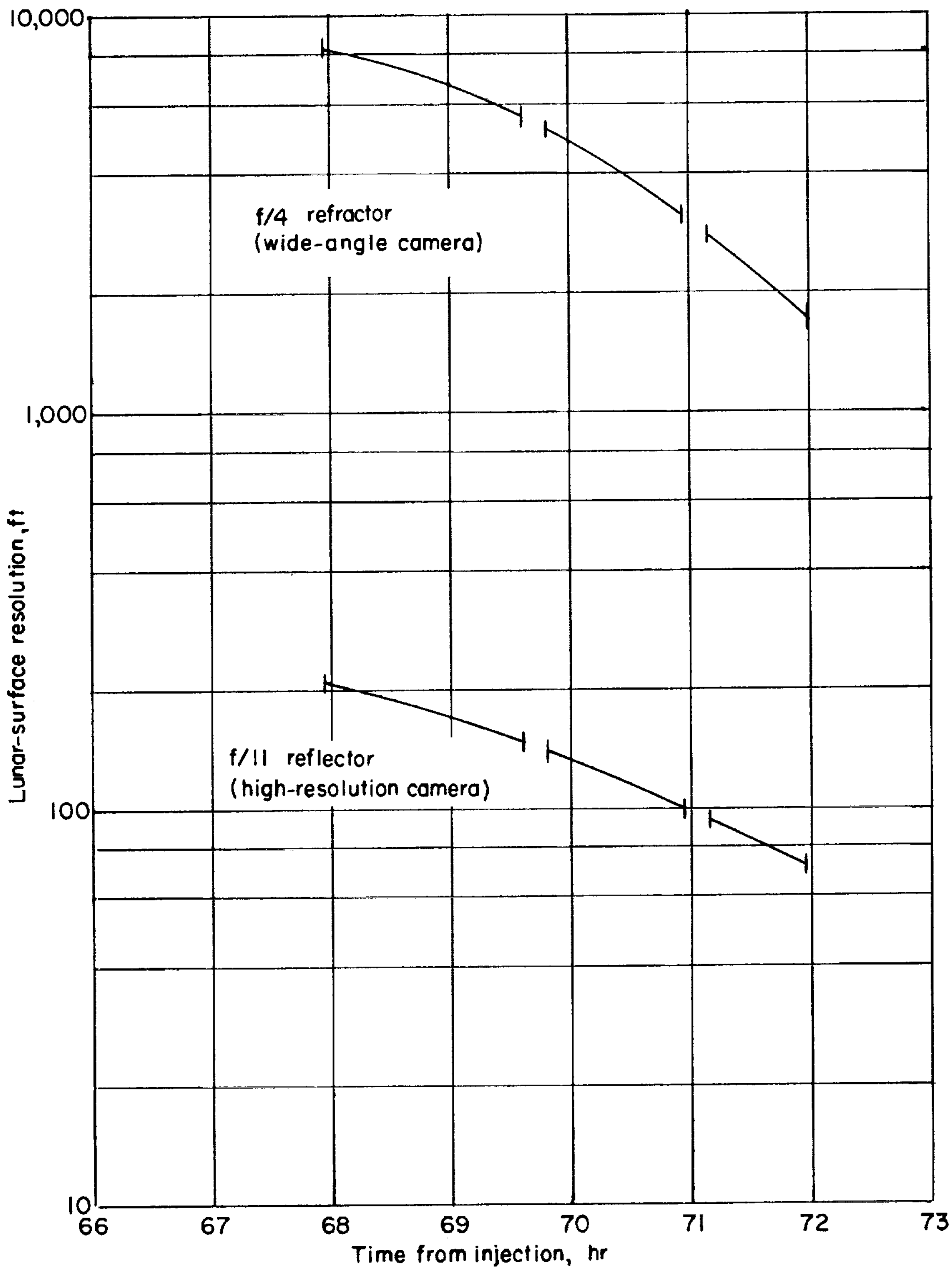

Figure 7.- Photographic resolution of lunar surface.

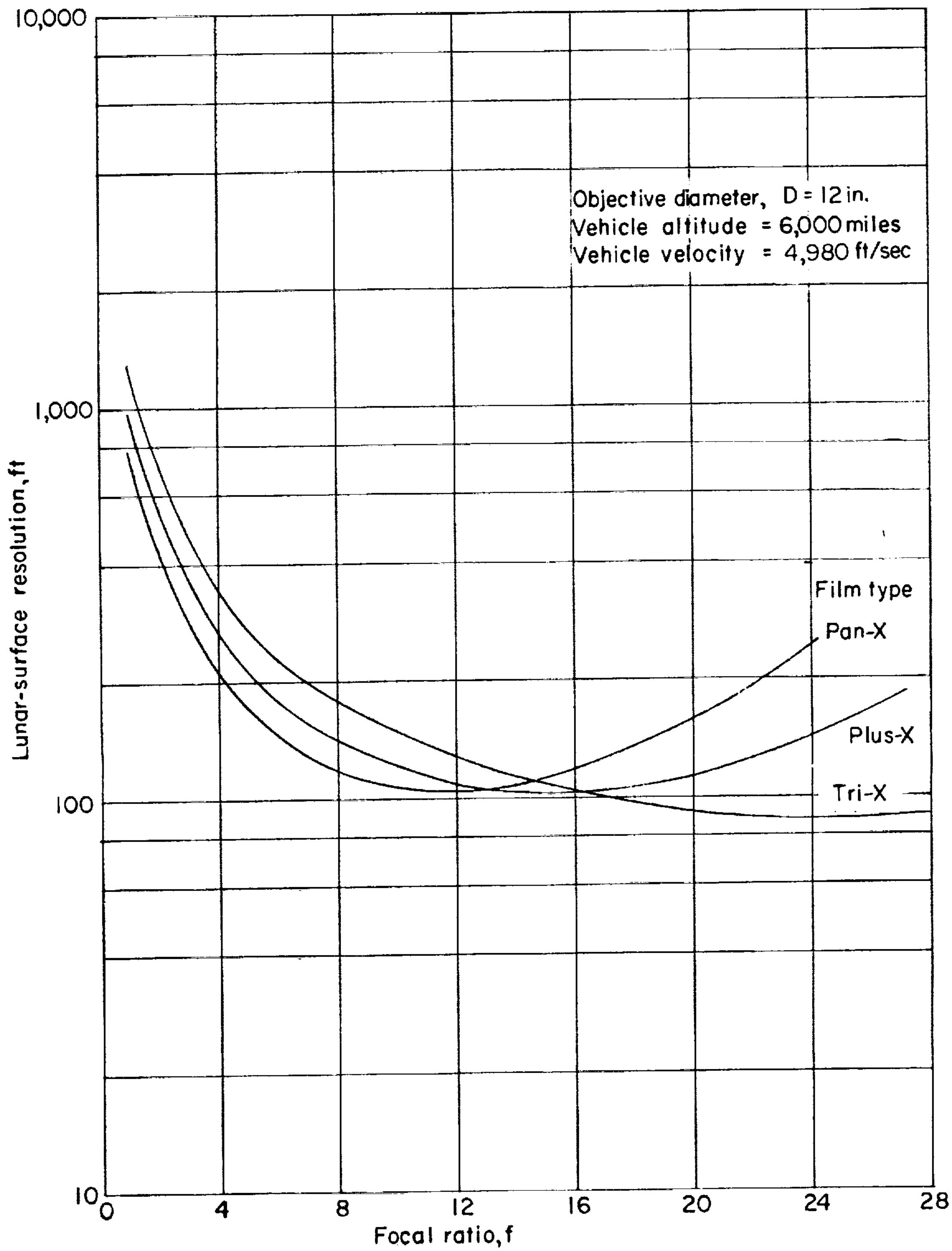

Figure 8.- Film properties for reflector system.

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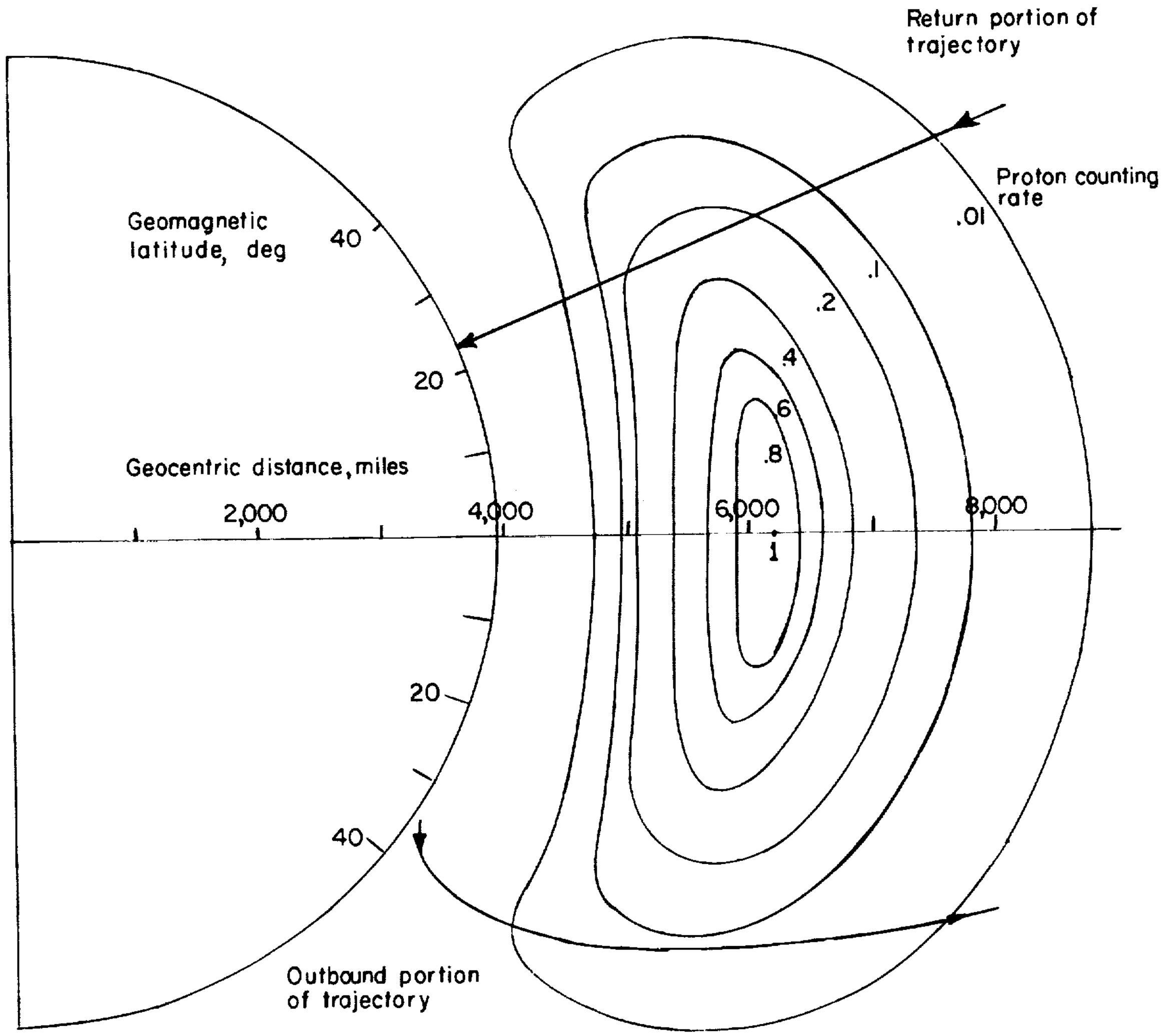

Figure 9.- Position of design trajectory with respect to inner Van Allen radiation belt.

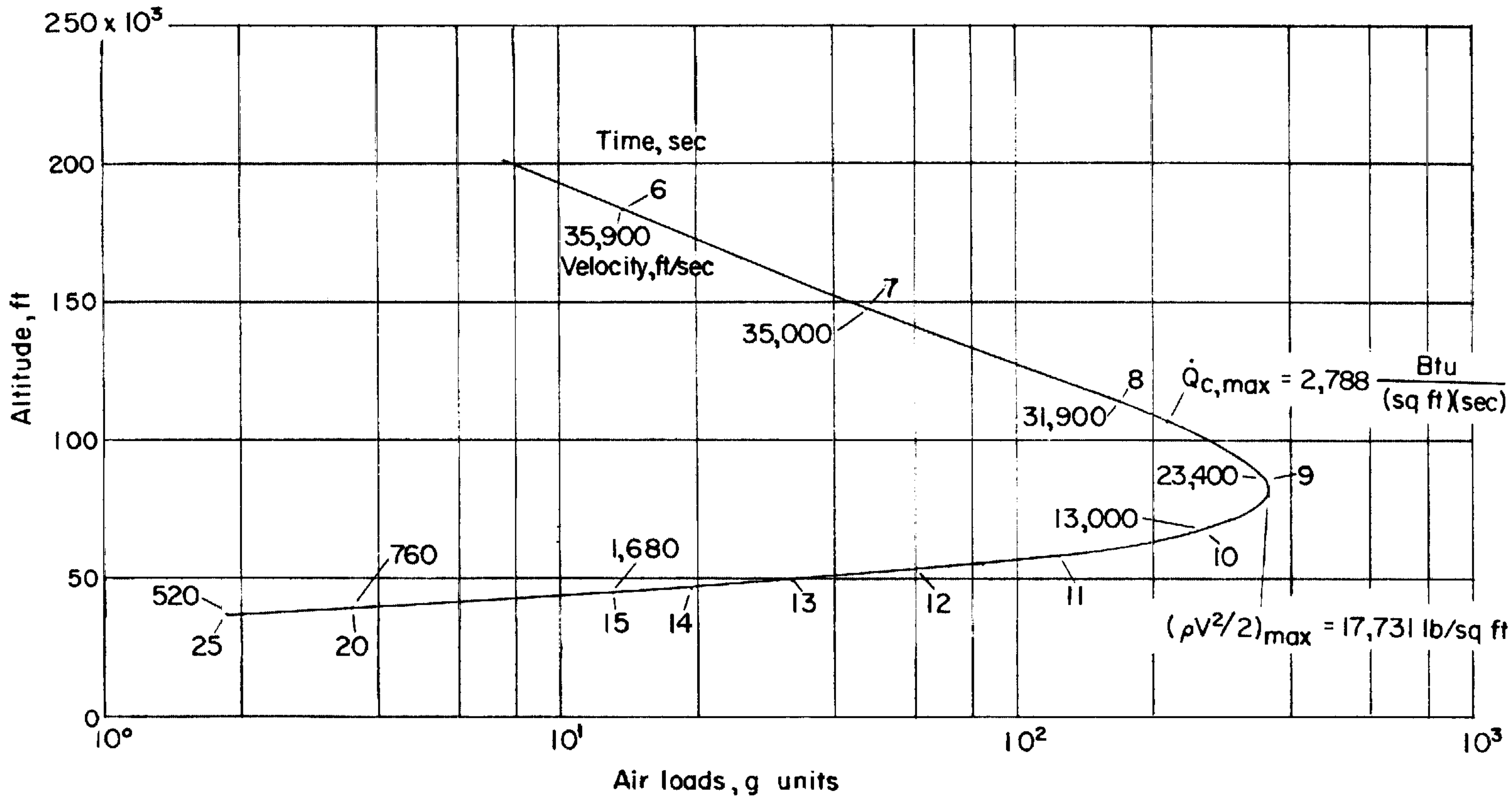

Figure 10.- Trajectory characteristics for vertical entry into atmosphere. Entry conditions: altitude = 400,000 ft or 76 miles; velocity = 36,000 ft/sec; W/C_DA = 50 lb/sq ft.

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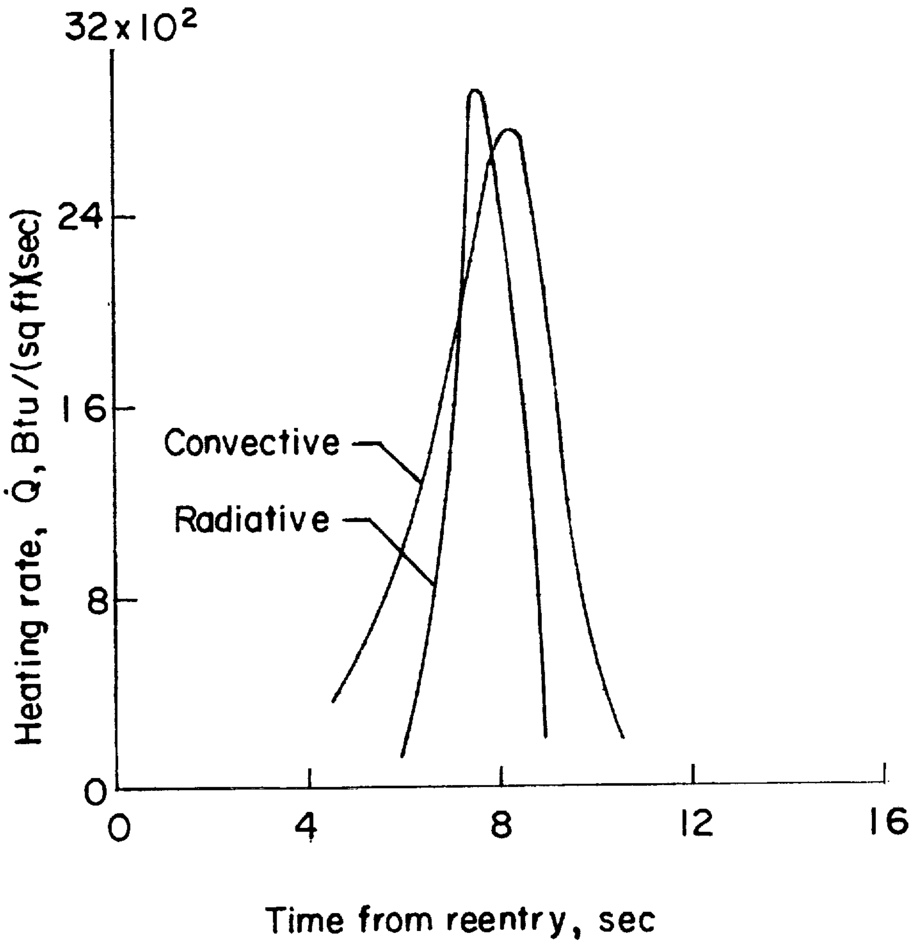

Figure 11.- Stagnation-point heat inputs. Nose radius = 1 ft; W/C_DA = 50 lb/sq ft.

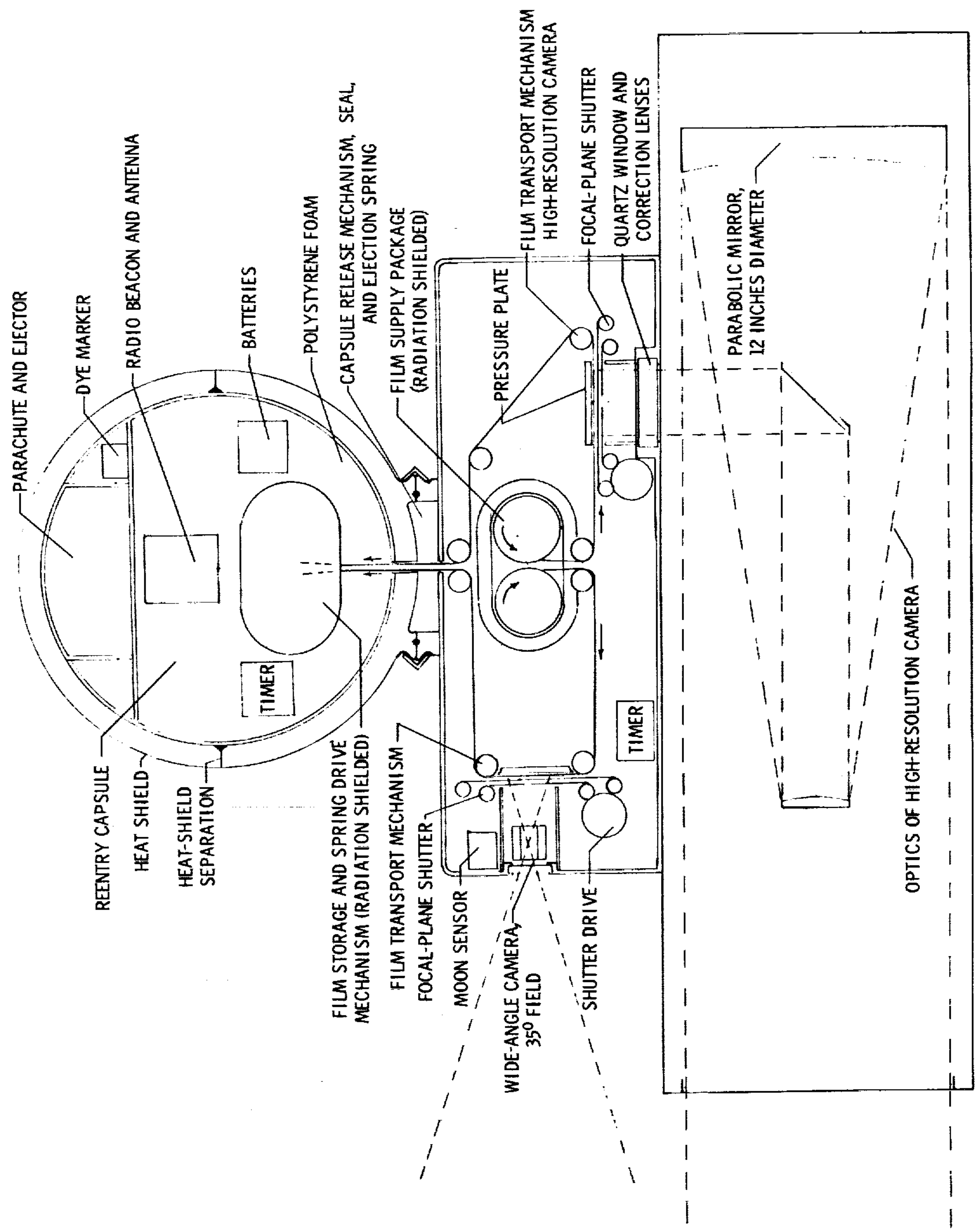

Figure 12.- Experimental package design.

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NASA TN D-1226
National Aeronautics and Space Administration.
FEASIBILITY STUDY OF A CIRCUMLUNAR PHOTOGRAPHIC EXPERIMENT. William H. Michael, Jr., Robert H. Tolson, and John P. Gapcynski. May 1962. 38p. OTS price, $1.00.
(NASA TECHNICAL NOTE D-1226)

A study has been made to investigate the feasibility of a high-resolution, lunar-surface photographic experiment, with the use of a circumlunar trajectory and with recovery of the film on return to the surface of the earth. Particular attention has been given to procedures for obtaining high-resolution photographs of the lunar surface, for returning the undeveloped film to the earth, and for recovering the data package on completion of such a mission. As an example of a typical existing vehicle which could be used for such an experiment, the characteristics of the Ranger spacecraft have been used in applicable portions of the study.

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